NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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TURBOJET THRUST AUGMENTATION BY EVAPORATION OF WATER

PRIOR TO MECHANICAL COMPRESSION AS DETERMINED

BY USE OF PSYCHROMETRIC CHART

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SUMMARY

The thrust augmentation of a turbojet engine resulting from the evaporation of water prior to mechanical compression was inves tigated over a range of flight Mach numbers, altitudes, ambient temperatures, and ambient relative humidities. The amount of cooling obtained by evaporation of the water was determined by means of a specially prepared psychrometric chart having total pressure as a variable. Both the augmented and normal engine performance were then evaluated for the various operating conditions by means of a generalized analysis of turbojet-engine performance.

The amount of thrust augmentation possible from compressor-inlet-air saturation increased with an increase in flight Mach number and decreased as the altitude was increased. For example, for an inlet-diffuser efficiency of 0.8, increasing the sea-level flight Mach number from 0 to 1.75 increased the ratio of augmented to normal thrust from 1.03 to 1.92. For a flight Mach number of 1.75, increasing the altitude from sea level to 35,332 feet reduced the ratio of augmented to normal thrust from 1.92 to 1.31.

INTRODUCTION

When water is injected into the compressor inlet of a turbo-jet engine, the evaporative cooling obtained results in increased engine thrust at the expense of increased liquid consumption. An analysis of the evaporative-cooling process and the attendant thrust augmentation can be conveniently divided into two phases: (1) cooling occurring at constant pressure before the air enters the compressor, and (2) the additional cooling associated with further evaporation of water during the mechanical-compression process. An analysis of evaporation that occurs during mechanical

compression after saturation of the inlet air and the thrust aug-
mentation produced by the injection of sufficient water at the
compressor inlet to saturate the compressor-outlet air is
reported in reference 1.

An analysis of the constant-pressure evaporation upstream of
the compressor and the attendant thrust augmentation was made at
the NACA Lewis laboratory. A specialized psychrometric chart that
permits calculation of the temperature resulting from the evapo-
ration of water upstream of the compressor is presented. This
psychrometric chart differs from the usual form, which is valid
for one pressure only, in that the temperature range has been
greatly extended and pressure is included as a variable. Also
presented is a method for calculating turbojet-engine perform-
ance over a range of altitudes and flight Mach numbers from data
obtained at any given flight condition. For purposes of illus-
tration, this generalized performance analysis is applied only in
the case of data obtained at sea-level, zero Mach number conditions.
The psychrometric chart and the generalized performance analysis
were used to calculate the thrust augmentation resulting from
water evaporation prior to compression for a typical turbojet
engine operating over a range of flight Mach numbers from 0 to
1.75 at altitudes of sea level and 35,332 feet. Also investi-
gated were the effects of variation in inlet-air temperature and
atmospheric humidity on thrust augmentation produced at sea-level,
zero flight Mach number conditions.

ANALYSIS

In evaluating the effect of compressor-inlet-air saturation
on engine performance, both the cooling due to the evaporation of
water and the effect of the resulting decreased inlet temperature
on engine performance must be determined. The amount of cooling
obtained was determined by means of the psychrometric chart pre-
sented herein. The effect of the reduced inlet-air temperatures
on engine performance could be determined from a conventional
thermodynamic analysis of the performance of the various engine
components by using assumed component efficiencies, as illustrated
in reference 1. In the present investigation, however, the deter-
mination of engine performance at various inlet-air temperatures
is based on a generalized method of presentation of turbojet-engine
performance in terms of the pumping characteristics of the engine.
(See reference 2.) By means of this analysis, it is possible to
describe both the augmented and normal engine performance over a
range of flight Mach numbers and altitudes from data obtained

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at sea-level, zero Mach number conditions. In order to indicate the thrust augmentation possible from saturation of the compressor-inlet air with water for a range of flight conditions, this generalized analysis is applied to performance data for a typical, current axial-flow turbojet engine.

Psychrometric Chart

The familiar psychrometric chart used in air-conditioning calculations is usually limited to standard sea-level total pressure and to temperatures less than 300° F. With changes in flight conditions, the pressures and the temperatures at the compressor inlet of a turbojet engine vary widely over ranges not included in the usual psychrometric chart; a chart was therefore constructed to include these ranges of pressure and temperature.

In deriving the psychrometric chart, the following assumptions were made:

(a) The air and water vapor are always at the same temperature.

(b) Dalton's law of partial pressures is valid.

(c) The temperature of the injected liquid water is 518.4° R.

(d) The thermodynamic properties of the water vapor are functions only of temperature for the range of vapor pressure considered.

By means of these assumptions, the theory of mixtures, and data for the thermodynamic properties of air and water vapor (references 3 and 4, respectively), a psychrometric chart was constructed with pressure as a variable and for temperatures up to 1500° R. The symbols used in the derivation of the chart are defined in appendix A; the details and the required equations are presented in appendix B.

Generalized Performance Analysis

When the compressor, burner, and turbine combination of a turbojet engine is considered as a pump and performance data are presented in terms of the pressure and temperature increases produced by this pump, it is possible to predict engine performance over a range of flight Mach numbers and altitudes from data

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obtained at sea-level, zero flight Mach number conditions and from
a knowledge of the inlet-diffuser and exhaust-nozzle performance.
The engine pressure ratio is defined as the ratio of turbine-
outlet total pressure to compressor-inlet total pressure; the
engine temperature ratio is defined as the ratio of turbine-outlet
total temperature to compressor-inlet total temperature. Air
flow, fuel flow, and engine pressure ratio are functions only of
the engine temperature ratio and the engine speed for given engine-
inlet conditions of temperature and pressure. The thrust and the
fuel consumption can therefore be determined for a range of flight
Mach numbers and altitudes from a knowledge of the desired flight
conditions and consequent engine-inlet temperature and pressure
and an assumed engine speed and turbine-outlet temperature.

Inasmuch as operation at rated engine speed at low inlet-air
temperatures involves corrected engine speeds that could not be
simulated at normal sea-level temperatures without dangerously
overspeeding the engine, the minimum flight Mach number for which
performance can be obtained by means of the generalized perform-
ance analysis is dependent on the altitude. At any altitude, the
minimum flight Mach number is that for which the compressor-inlet
total temperature is equal to the compressor-inlet total tempera-
ture for the sea-level, zero flight Mach number investigations.
For data obtained at standard sea-level temperature, the minimum
flight Mach number increases from 0.6 at an altitude of 10,000 feet
to 1.25 at 35,332 feet. This minimum flight Mach number limita-
tion could be avoided by using data for which refrigerated inlet-
air was supplied to the engine.

The accuracy of the generalized performance analysis when used
in calculating performance at various altitudes is dependent on
the degree to which the familiar corrected engine performance
parameters can be generalized for various altitudes. The great-
est errors involved probably occur in fuel consumption, where the
combustion efficiency changes markedly with altitude. Changes in
component efficiencies resulting from changes in flight Mach num-
ber are accounted for by this method of analysis.

The details of the generalized performance analysis and the
performance obtained for a typical engine are presented in appen-
dix C. A discussion of other applications of the pumping char-
acteristics of turbojet engines is presented in reference 2.

Augmentation Calculations

By use of the psychrometric chart to evaluate cooling due to evaporation of water and the generalized performance analysis to obtain normal and augmented engine performance, the thrust augmentation resulting from water evaporation before compression was determined for a typical turbojet engine. The curves presented are for constant engine speed and turbine-outlet temperature and require that the engine be equipped with a variable-area exhaust nozzle.

No account was taken of the effect of the change in thermodynamic properties of the working fluid produced by the addition of water vapor on the performance curves for the engine. Experimental results indicate that water content does have some effect on these curves; however, the effect is relatively slight and no completely satisfactory method has, as yet, been devised to account for it.

RESULTS AND DISCUSSION

Psychrometric Chart

In the psychrometric chart presented in figure 1, total enthalpy H is plotted against dry-bulb temperature t for various values of water-air ratio X. Also included are curves of the ratio of relative humidity to relative pressure φ/δ, where relative pressure δ is defined as the ratio of static pressure to standard sea-level static pressure. Inasmuch as evaporation occurs at constant pressure, the value of δ is unchanged by water injection. The curves of constant water-air ratio X are approximately straight lines having a positive slope, and the curves of constant ratio of relative humidity to relative pressure φ/δ are curved upward. The distance between the lines of constant water-air ratio decreases as the water-air ratio increases. The enthalpy scale was arbitrarily chosen such that the enthalpy of a saturated mixture of air and water vapor at a temperature of 518.4° R and a pressure of 14.696 pounds per square inch was 100 Btu per pound of air.

The temperature resulting after water evaporation may be obtained from figure 1 for a wide range of initial air temperatures and pressures. The temperature can be found either for the evaporation of a given amount of water or for any final relative humidity. The method of using the chart is indicated by the following illustrative cases:

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Example (1). - For initially dry air at a temperature of
1260° R assume that sufficient water is injected to give a final
water-air ratio of 0.05. From figure 1, the total enthalpy is
271.6 Btu per pound of air for a temperature of 1260° R and a
water-air ratio of 0. Inasmuch as evaporation occurs at constant
total enthalpy, the temperature after evaporation can be found
from the initial total enthalpy and the final water-air ratio.
For a total enthalpy of 271.6 Btu per pound of air and a water-
air ratio of 0.05, the temperature is 1008° R. The temperature
drop is then 252° for these particular conditions.

Example (2). - In order to illustrate the use of the chart
in obtaining the temperature and the water-air ratio necessary to
saturate air from a given set of initial conditions, initially dry
air at a temperature of 1060° R and at twice standard sea-level
static pressure is assumed. The enthalpy corresponding to a tem-
perature of 1060° R and a water-air ratio of 0 is 221.0 Btu per pound
of air. The value of relative humidity for saturation is 1.0 and
for these conditions the value of δ is 2.0. The temperature and
water-air ratios are 614° R and 0.0995, respectively, at an
enthalpy of 221.0 Btu per pound of air and a value of φ/δ of 0.5.
For these conditions, the initial mixture may be cooled 446° by
injecting 0.0995 pound of water per pound of air.

Example (3). - The use of the chart in obtaining temperatures
resulting from water evaporation when some water is initially
present in the air is the same as that previously described,
except that the starting enthalpy is found from the initial water-
air ratio and temperature. For example, for the conditions of
temperature and pressure in example (2) and an initial water-air
ratio of 0.02, the initial enthalpy is 247.0 Btu per pound of air.
The saturation temperature for an enthalpy of 247.0 Btu per pound
and a value of φ/δ of 0.5 is 620° R and the water-air ratio
required is 0.122. For these conditions, cooling the initial mix-
ture 440° is therefore possible by evaporating 0.102 pound of water
per pound of air.

The effect of initial temperature and pressure on the amount
of cooling possible from saturation can be readily seen from fig-
ure 1. For a constant pressure and consequent constant value of
φ/δ ($\varphi = 1$), the amount of cooling possible increases as the
initial temperature is raised. For a constant temperature,
increasing the pressure decreases the value of φ/δ, and from
figure 1 decreasing the value of φ/δ is seen to decrease the
amount of cooling obtained. In general, when water is initially
present, the cooling possible from saturation is somewhat less

than for dry air, and the amount of water that must be injected to obtain saturation may be either somewhat greater or less, depending on the initial conditions.

In the construction of figure 1, it has been assumed that the enthalpy of liquid water is zero at 518.4° R and that the water injected for cooling is at this temperature. For water at temperatures only slightly different from 518.4° R, the errors in temperature and water-air ratio involved will be negligible; in extreme cases, however, the error in the amount of water required for a given amount of cooling may become appreciable. For example, figure 1 indicates that initially dry air at a temperature of 910° R and standard sea-level pressure may be cooled 333° by saturation with water. The water-air ratio required as indicated by figure 1 is 0.075. For water injected at 618.4° R rather than 518.4° R, the amount of cooling possible, as determined by the method discussed in appendix B, would be 331°; the required water-air ratio would be 0.081. For temperatures of injected water less than 518.4° R, the chart would predict slightly less cooling than actually possible and somewhat greater water-air ratios than actually necessary. For partial saturation, these differences would be reduced.

Generalized Performance Analysis

The curves necessary for predicting performance at various flight Mach numbers and altitudes from experimental data obtained at sea-level, zero Mach number conditions are presented in figure 2. These curves were obtained from the performance of a current turbojet engine having an axial-flow compressor and developing 4000 pounds thrust at sea-level, zero Mach number conditions.

The method of obtaining these curves from the usual turbojet-engine performance quantities is presented in appendix C.

In figure 2(a) the engine pressure ratio P_7/P_1, the ratio of total pressure at turbine outlet to total pressure at compressor inlet, is shown as a function of inlet-temperature factor $\theta(7700/N)^2$ for various values of the factor $(T_7/1660)(7700/N)^2$. The term θ is defined as the ratio of compressor-inlet total temperature to standard sea-level static temperature, N is the engine speed, T_7 is the turbine-outlet total temperature, and the constants 7700 and 1660 are rated engine speed and rated turbine-outlet total temperature, respectively, for the particular engine considered.

The factor $\theta(7700/N)^2$ is readily obtainable from the more famil-
iar corrected-engine speed $N/\sqrt{\theta}$. (The term 7700 is included to
allow the abscissa of figure 2 to be read directly in terms of θ
for the condition of rated engine speed.) Curves are included for
values of $(T_7/1660)(7700/N)^2$ of 0.9, 1.0, and 1.1. All con-
ditions for which the engine is operating at rated turbine-outlet
temperature and rated engine speed are presented by the curve for
which the value of $(T_7/1660)(7700/N)^2$ is 1.0. Operating con-
ditions other than design speed and turbine-outlet temperature
are represented by the curves for which the value of
$(T_7/1660)(7700/N)^2$ is different from 1.0. It can be seen from
figure 2(a) that for a constant value of $(T_7/1660)(7700/N)^2$, the
engine pressure ratio decreases as θ increases; hence, the
engine pressure ratio decreases as the flight Mach number is
increased and increases as the altitude is increased.

In figures 2(b) and 2(c), the engine total-gas-flow fac-
tor $(W_g/\delta)(N/7700)$ and fuel-flow factor $(W_f/\delta)(7700/N)$, respec-
tively, are shown as functions of the inlet-temperature fac-
tor $\theta(7700/N)^2$ for various values of the factor $(T_7/1660)(7700/N)^2$.
The gas-flow factor $(W_g/\delta)(N/7700)$ is obtained by taking the sum
of the air flow and the fuel flow and correcting the sum in the
same manner as air flow. Because air flow and the fuel flow
should not actually be corrected in the same manner, this factor
is a somewhat inaccurate measure of the corrected total gas flow;
however, fuel flow is a small part of the total gas flow and no
serious inaccuracy is thus involved.

The gas-flow and fuel-flow factors (figs. 2(b) and 2(c)) can
be readily obtained from the more familiar corrected air-flow and
fuel-flow parameters $W_a\sqrt{\theta}/\delta$ and $W_f/(\delta\sqrt{\theta})$, respectively. In
obtaining performance data at high flight speeds, the effect of
varying inlet conditions for constant engine speed and turbine-
outlet temperature must be determined; in figure 2, the usual
performance parameters have therefore been so altered that the
inlet-temperature parameter θ becomes the primary variable and
the engine speed N, the correction parameter.

In order to simplify the computations of performance at vari-
ous flight Mach numbers by means of the generalized performance
analysis, curves of inlet-diffuser and exhaust-nozzle performance
are presented in figures 3 and 4, respectively. The inlet-diffuser
pressure ratio δ/δ_0 for various values of diffuser adiabatic
efficiency is shown in figure 3(a) as a function of flight Mach
number. The inlet-diffuser temperature ratio θ/θ_0, which is

independent of diffuser efficiency, is shown as a function of flight Mach number in figure 3(b). The terms δ_0 and θ_0 have been included in figure 3 in order to generalize the curves for all altitudes.

The effective exhaust-jet velocity $V_j\sqrt{1660/T_7}$ is shown as a function of exhaust-nozzle pressure ratio P_7/p_0 for various values of exhaust-nozzle adiabatic efficiency in figure 4. The constant 1660 has been introduced in order to allow the curves of figure 4 to be read directly in terms of jet velocity for a turbine-outlet temperature of 1660° R, which is the normal value for the engine used to illustrate the generalized performance analysis. The curves of figure 4 are for a simple convergent-type nozzle and include the thrust contribution of the excess pressure at the nozzle throat.

The method of using the curves of figures 2 to 4 to obtain engine performance is illustrated by means of the following example:

Assume that the engine is operating at an engine speed of 7700 rpm and that the exhaust-nozzle area is variable so that a turbine-outlet temperature of 1660° R can be maintained. Further assume that performance at a sea-level Mach number of 1.0 is desired and that the engine is equipped with an inlet diffuser and an exhaust nozzle having adiabatic efficiencies of 0.85 and 0.96, respectively, at these conditions. The performance can then be evaluated by the following steps:

1. For sea level $(\delta_0 = 1, \quad \theta_0 = 1)$ at a flight Mach number of 1.0 and a diffuser adiabatic efficiency of 0.85, the total-pressure ratio across the inlet diffuser δ/δ_0 from figure 3(a) is 1.74.

2. For the same conditions, the inlet-diffuser total-temperature ratio θ/θ_0 from figure 3(b) is 1.2.

3. For an engine speed of 7700 rpm, a turbine-outlet temperature of 1660° R, and a value of θ of 1.2, the following quantities can be determined from figure 2: engine pressure ratio P_7/P_1 is 1.496 (fig. 2(a)); gas-flow factor $(W_g/\delta)(N/7700)$ is 60.8 (fig. 2(b)); and the fuel-flow factor $(W_f/\delta)(7700/N)$ is 0.870 (fig. 2(c)).

4. The total gas flow W_g and the fuel flow W_f are found, from the value of δ and the corrected gas-flow and fuel-flow factors, to be 105.6 and 1.514 pounds per second, respectively.

5. The exhaust-nozzle pressure ratio P_7/p_0 is obtained from the product of the diffuser pressure ratio δ/δ_0 and the engine pressure ratio P_7/P_1; the value of P_7/p_0 is found to be 2.603.

6. For the given values of exhaust-nozzle efficiency η_n, turbine-outlet temperature T_7, and the calculated exhaust-nozzle pressure ratio P_7/p_0, the effective jet velocity V_j is found, from figure 4, to be 2180 feet per second.

7. For a sea-level flight Mach number of 1.0, the flight velocity V_0 is 1117 feet per second.

8. By using the jet velocity V_j, the flight velocity V_0, the gas flow W_g, and the fuel flow W_f, the net thrust F of the engine is found from equation (C1) of appendix C to be 3538 pounds.

9. The specific fuel consumption is, from the thrust and the fuel flow, 1.541 pounds per hour per pound of thrust.

Augmentation Calculations

By using the psychrometric chart (fig. 1) to evaluate inlet temperatures for saturation and the previously described generalized performance analysis to evaluate both normal and augmented engine performance, the thrust augmentation resulting from compressor-inlet saturation was determined for the engine discussed herein. In all cases it was assumed that the engine speed and the turbine-outlet temperature were maintained at the rated values. It was also assumed that the inlet air was diffused to a sufficiently low velocity such that the static temperature did not differ appreciably from the total temperature at the compressor inlet.

The augmented-thrust ratio as a function of flight Mach number for values of inlet-diffuser adiabatic efficiency of 0.6, 0.8, and 1.0 and for altitudes of sea level and 35,332 feet is shown in figure 5. In calculating the data presented in figure 5, the atmospheric relative humidity was assumed to be 0.50 and, for each condition, sufficient water was assumed to be injected at the compressor inlet to saturate the air at this point.

Increasing the flight Mach number increases the augmented-thrust ratio (fig. 5). For sea-level altitude and an inlet-diffuser adiabatic efficiency of 0.8, the augmented-thrust ratio increases from 1.03 at a flight Mach number of 0 to 1.92 at a flight Mach

number of 1.75. The increased thrust augmentation at increased
flight Mach numbers is a result of the greater amount of cooling
possible at high inlet temperatures. For a flight Mach number
of 1.75 and diffuser adiabatic efficiency of 0.8, increasing the
altitude from sea level to 35,332 feet decreases the augmented-
thrust ratio from 1.92 to 1.31. This decreased augmented-thrust
ratio at the higher altitude is a result of the decreased cooling
possible at the low temperatures associated with high altitudes.
Although the actual augmented thrust will be lower for lower
values of inlet-diffuser efficiency, the augmented-thrust ratio
increases for lowered values of diffuser efficiency because of
the decreased performance of the normal engine.

The top curves of figure 5 give the augmented-liquid ratio
corresponding to the augmented-thrust ratios shown in the lower
curves. The total liquid consumption for the augmented engine
includes both fuel and injected water, whereas that of the normal
engine is for fuel alone. The augmented-liquid ratio increases
rapidly as the flight Mach number is increased and decreases as
the altitude is increased from sea level to 35,332 feet. For
example, increasing the sea-level flight Mach number from 0 to
1.75 for a diffuser efficiency of 0.8 results in an increase in
augmented-liquid ratio from 1.15 to 7.6. The increased liquid
consumptions at increased flight Mach numbers and the decreased
liquid consumption at high altitudes are a result of the effect
of inlet temperature on the amount of water that may be evaporated,
as was previously mentioned. The augmented-liquid ratio is some-
what higher for lower diffuser efficiencies because the decreased
pressures at the compressor inlet allow the evaporation of more
water.

•The curves presented assume complete evaporation of the
injected water prior to mechanical compression; however, the val-
ues of thrust augmentation shown would not be greatly influenced
by incomplete evaporation, inasmuch as the water would later
evaporate in the compressor.

The effect of atmospheric relative humidity on the augmented-
thrust ratio is shown in figure 6. The augmented-thrust ratio is
shown as a function of flight Mach number for values of atmos-
pheric relative humidity of 0, 0.50, and 1.00. These curves are
for sea-level altitude and an inlet-diffuser efficiency of 0.8.
The relative humidity of the atmosphere has a small effect on
the augmented-thrust ratio, with the greatest effect occurring
at low flight Mach numbers. A change in atmospheric relative

humidity from 0 to 1.00 changes the augmented-thrust ratio from
1.07 to 1.00 at a flight Mach number of 0 and from 1.95 to 1.90
for a flight Mach number of 1.75.

The thrust augmentation at standard sea-level, zero Mach num-
ber conditions is very low, as indicated in figure 5. The data
of figure 5 are for an atmospheric temperature of 59° F; for tem-
peratures greater than 59° F, however, more water can be evaporated
and the thrust increase will be greater.

This effect is indicated in figure 7, wherein the augmented-
thrust ratio is shown as a function of inlet temperature for vari-
ous values of atmospheric relative humidity at standard sea-level
pressure and zero flight Mach number. For an atmospheric rela-
tive humidity of 1.00, no thrust increase can be obtained, inas-
much as no evaporative cooling is possible; for approximately dry
air, however, appreciable gains in thrust are possible, with
higher temperatures resulting in higher thrust increases. For
example, at a temperature of 120° F, the augmented-thrust ratio
for an atmospheric relative humidity of 0 is 1.21 as compared
with 1.07 at a temperature of 59° F. Although the augmented-
thrust ratio is greater at increased temperatures, the actual
thrust will be lowered because of the marked decrease in normal
thrust at the lower corrected engine speeds accompanying the
increased temperatures.

SUMMARY OF RESULTS

The following results were obtained from an investigation
of the thrust augmentation of turbojet engines by the injection
of sufficient water at the compressor inlet to saturate the
compressor-inlet air:

1. The augmented-thrust ratio theoretically possible from
compressor-inlet saturation increased rapidly as the flight Mach
number was increased and decreased as the altitude was increased.
For an atmospheric relative humidity of 0.50 and an inlet-diffuser
adiabatic efficiency of 0.8, increasing the sea-level flight Mach
number from 0 to 1.75 increased the augmented-thrust ratio from
1.03 to 1.92. The augmented-liquid ratio for the 1.75 Mach number
condition was 7.6. For a flight Mach number of 1.75, increasing
the altitude from sea level to 35,332 feet reduced the augmented-
thrust ratio from 1.92 to 1.31.

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2. For standard atmospheric temperatures, the relative humidity of the atmosphere had a small effect on the augmented-thrust ratio produced by compressor-inlet saturation at all flight speeds.

3. For sea-level, zero Mach number conditions and low atmospheric relative humidities, the augmented-thrust ratio increased as the inlet-air temperature was increased. For high atmospheric relative humidities the effect of inlet temperature was very small.

Lewis Flight Propulsion Laboratory,
    National Advisory Committee for Aeronautics,
        Cleveland, Ohio, December 2, 1949.

APPENDIX A

SYMBOLS

The following symbols are used in this report:

F net thrust, lb

g acceleration due to gravity, ft/sec^2

H total enthalpy of air and water vapor, Btu/lb-air

h enthalpy, Btu/lb

M_O flight Mach number

N engine speed, rpm

P total pressure, lb/sq in.

p static or partial pressure, lb/sq in.

p_t sum of partial pressures of air and water vapor, lb/sq in.

$p_{v,s}$ partial pressure of water vapor at saturation, lb/sq in.

R gas constant, ft-lb/(lb)(^{O}R)

T total temperature, ^{O}R

t static temperature, ^{O}R

V velocity, ft/sec

v specific volume, cu ft/lb

W weight flow, lb/sec

X water-air ratio, lb/lb

γ ratio of specific heats

δ ratio of compressor-inlet total pressure to standard sea-level static pressure, $P_1/14.696$

δ_0 ratio of ambient static pressure to standard sea-level static pressure, $p_0/14.696$

η_n exhaust-nozzle adiabatic efficiency

θ ratio of compressor-inlet total temperature to standard sea-level static temperature, $T_1/518.4$

θ_0 ratio of ambient static temperature to standard sea-level static temperature, $t_0/518.4$

ρ density, lb/cu ft

φ relative humidity, $p_v/p_{v,s}$

Subscripts:

a   air

b   conditions before evaporation

c   conditions after evaporation

f   fuel

g   total gas (air plus fuel plus water)

l liquid water

j   exhaust jet

v   water vapor

0   free-stream conditions

1   compressor-inlet conditions

7   turbine-outlet conditions

APPENDIX B

DERIVATION OF PSYCHROMETRIC CHART

The equations necessary for obtaining a psychrometric chart having total pressure as a variable may be derived in the following manner:

From the general energy equation, the enthalpy of a mixture of air and water per pound of air is

$$h_{a,b} + X_b h_{v,b} + (X_c - X_b) h_l = h_{a,c} + X_c h_{v,c} \tag{B1}$$

The enthalpy of liquid water is small compared to the enthalpy of water vapor; by assuming that the water is injected at the temperature for which the enthalpy is zero, the effect of liquid water may be neglected. For the present analysis it was assumed that the enthalpy of liquid water was zero at 518.4° R; and the results are therefore accurate for water injected at this temperature, with very small errors resulting for water at a somewhat different temperature. By assuming that the enthalpy of liquid water is zero, equation (B1) may be stated as

$$h_{a,b} + X_b h_{v,b} = h_{a,c} + X_c h_{v,c} \tag{B2}$$

The total enthalpy H for condition b or c is defined as

$$H = h_a + X h_v \tag{B3}$$

The water-air ratio X is evaluated as follows:

$$X = \frac{\rho_v}{\rho_a} = \frac{p_v}{R_v t_v} \frac{R_a t_a}{p_a} \tag{B4}$$

If the temperatures of the air and the vaporized water are equal and Dalton's law of partial pressures is valid, equation (B4) becomes

$$X = \frac{R_a}{R_v} \frac{p_v}{p_t - p_v} \tag{B5}$$

From the definitions of relative humidity, relative pressure, and gas constant

$$\varphi = p_V/p_{V,s} \tag{B6}$$

$$\delta = p_t/14.696 \tag{B7}$$

$$R_V = \frac{p_V v_V}{t_V} \tag{B8}$$

Substituting equations (B6) to (B8) in equation (B5) gives

$$X = \frac{R_a}{\left(\dfrac{p_V v_V}{t_V}\right)}\left(\frac{\dfrac{\varphi}{\delta}\dfrac{p_{V,s}}{14.696}}{1 - \dfrac{\varphi}{\delta}\dfrac{p_{V,s}}{14.696}}\right) \tag{B9}$$

When water-air ratio X is replaced in equation (B3) by the value as expressed in equation (B9), the expression for total enthalpy H becomes

$$H = h_a + h_V \frac{R_a}{\left(\dfrac{p_V v_V}{t_V}\right)}\left(\frac{\dfrac{\varphi}{\delta}\dfrac{p_{V,s}}{14.696}}{1 - \dfrac{\varphi}{\delta}\dfrac{p_{V,s}}{14.696}}\right) \tag{B10}$$

The data presented in figure 1 were obtained by means of equations (B9) and (B10) and the thermodynamic data for air and water vapor contained in references 3 and 4, respectively. For convenience, the enthalpy of the saturated mixture of air and water vapor at a temperature of 518.4° R and a pressure of 14.696 pounds per square inch was arbitrarily fixed at 100 Btu per pound of air. In using equation (B10), the enthalpies of the air and water vapor obtained from references 3 and 4, respectively, must be altered in order to satisfy the conditions of zero enthalpy of liquid water at 518.4° R and the condition of an enthalpy of 100 Btu per pound of air for a saturated mixture of air and water vapor at a temperature of 518.4° R and a pressure of 14.696 pounds per square inch. The values of enthalpy of water vapor must be decreased by 27.1 Btu per pound in order to satisfy the first condition and the enthalpy of air must be increased by 60.25 Btu per pound in order to satisfy the second

condition. The injection of water at a temperature other than
518.4° R would simply alter the values of enthalpy of water vapor
and thereby change the total enthalpy of the mixture for a given
temperature.

The enthalpy and the specific volume of water vapor are func-
tions of the vapor pressure of water as well as the temperature.
In the range of temperatures and pressures encountered at the com-
pressor inlet of a turbojet engine, however, the enthalpy of
water vapor and the product of vapor pressure and specific volume
can be considered as functions only of temperature for all prac-
tical purposes. For any value of temperature, the enthalpy and
the specific volume were evaluated for the maximum pressure that
would exist for that temperature at the compressor inlet of a
turbojet engine. This maximum pressure is the pressure that would
be obtained by ideal ram compression from standard sea-level tem-
perature and pressure to the desired temperature.

APPENDIX C

DEVELOPMENT OF GENERALIZED PERFORMANCE ANALYSIS

The usual method of presenting turbojet-engine data is to show corrected thrust F/δ, corrected air flow $W_a\sqrt{\theta}/\delta$, corrected fuel flow $W_f/(\delta\sqrt{\theta})$, and corrected turbine-outlet temperature T_7/θ as functions of corrected engine speed $N/\sqrt{\theta}$. The accuracy of the generalized performance analysis in calculating performance for various altitudes is dependent on the degree to which these engine performance parameters may be generalized for various altitudes. Experimental results indicate that the thrust, the air flow, and the turbine-outlet temperature generalize very well for all altitudes, and that the greatest deviation may be expected for fuel flow because of the variation of combustion efficiency with altitude.

The effect of flight Mach number M_0 cannot be determined merely by considering the variation in θ and δ produced by a change in flight Mach number, because for a given corrected engine speed the pressure and temperature ratios across the engine change with changes in flight Mach number. By operating the engine at sea-level, zero Mach number conditions over a range of engine speeds and with several different exhaust-nozzle areas, it is possible to vary the engine temperature and pressure ratios over a range such as would be encountered by operation of the engine over a wide range of flight conditions. The usual method of presenting turbojet-engine data is to consider engine speed the primary variable and inlet temperature and pressure as correction factors. Because turbojet engines are usually operated very close to rated engine speed, the familiar performance parameters have been altered herein to make inlet temperature the primary variable and engine speed and inlet pressure the correction factors.

From the expression for net thrust developed by a turbojet engine,

$$F = \frac{W_g}{g}(V_j - V_0) + \frac{W_f}{g}V_0 \qquad (C1)$$

it can be seen that in order to evaluate the net thrust at any flight condition the gas flow through the engine, the fuel flow, the flight velocity, and the jet velocity must be known.

For an operating condition of known turbine-outlet temperature and engine speed, the gas flow through the engine and the fuel flow are functions only of the compressor-inlet temperature and pressure and may therefore be evaluated for any flight condition and inlet-diffuser performance.

The exhaust-jet velocity is a function of the exhaust-nozzle-inlet temperature, the pressure ratio across the exhaust nozzle, and the exhaust-nozzle efficiency. The exhaust-nozzle pressure ratio P_7/p_0 can be obtained from the engine pressure ratio P_7/P_1 and the diffuser pressure ratio P_1/p_0 as follows:

$$P_7/p_0 = (P_1/p_0)\,(P_7/P_1) \tag{C2}$$

The diffuser pressure ratio can be found either from a knowledge of the actual diffuser performance or calculated from an assumed diffuser efficiency and flight velocity. The engine pressure ratio is obtained during the sea-level, zero Mach number runs either by direct measurement or by calculation. The following expression may be used to compute the engine pressure ratio from data obtained at zero Mach number when thrust, air flow, fuel flow, and turbine-outlet temperature are known:

$$P_7/P_1 = \left[1 - \frac{1}{\eta_n}\,\frac{\gamma-1}{2g\gamma R}\,\frac{1}{T_7/\theta}\left(\frac{g\,F/\delta}{W_a\,\sqrt{\theta}/\delta + W_f\,\sqrt{\theta}/\delta}\right)^2\right]^{\frac{\gamma}{1-\gamma}} \tag{C3}$$

From the gas flow, fuel flow, jet velocity, and flight velocity, the net thrust produced by the turbojet engine may then be evaluated over a wide range of flight conditions by use of equation (C1).

The curves of figure 2 are obtained by cross-plotting the turbojet-engine performance data presented in figure 8. In figure 8(a) the engine pressure ratio P_7/P_1 calculated from equation (C3) for a constant exhaust-nozzle efficiency is shown as a function of the inlet-temperature factor $\theta(7700/N)^2$ for various values of exhaust-nozzle diameter. The factor $\theta(7700/N)^2$ is proportional to the reciprocal of the square of the more familiar parameter $N/\sqrt{\theta}$. The constant 7700 was introduced to allow the abscissa of figure 8 to be read directly in terms of θ for rated speed. In figures 8(b) and 8(c), the engine gas flow $(W_g/\delta)(N/7700)$ and fuel flow $(W_f/\delta)(7700/N)$, respectively, are presented as functions of $\theta(7700/N)^2$ for various values of exhaust-nozzle diameter.

The constant 7700 was introduced to obtain W_g/δ and W_f/δ directly from figures 8(b) and 8(c), respectively, for rated engine speed. The gas-flow factor $(W_g/\delta)(N/7700)$ is somewhat in error because it assumes that air flow and fuel flow are corrected in the same manner; however, fuel flow is a small part of the total gas flow and the gas-flow factor is sufficiently accurate for most purposes.

In figure 8(d), the reciprocal of turbine-outlet temperature $\theta(1660/T_7)$ is shown as a function of inlet-temperature factor $\theta(7700/N)^2$ for various exhaust-nozzle diameters. The factor 1660, turbine-outlet temperature for operation at rated speed, was included to allow the ordinate of figure 8(d) to be read directly in terms of θ. For all conditions of engine operation, the value of θ on the abscissa must be equal to the value of θ on the ordinate. For operation at rated engine speed and turbine-outlet temperature, the factor $(T_7/1660)(7700/N)^2$ must be equal to 1. In order to represent engine operation at other than design speed or turbine-outlet temperature, curves for values of the factor $(T_7/1660)(7700/N)^2$ of 0.9 and 1.1 are also included. The curves presented in figure 2 are obtained by cross-plotting the curves of figures 8(a) to 8(c) with figure 8(d). The data presented in figures 2 and 8 are for a typical 4000-pound thrust axial-flow engine; the same method of generalized performance analysis, however, can be applied to any engine for which sea-level, zero Mach number data for operation with various exhaust-nozzle areas are available.

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2. Sanders, Newell, and Behun, Michael: Generalization of Turbojet-Engine Performance in Terms of Pumping Characteristics. NACA TN 1927, 1949.

3. Keenan, Joseph H., and Kaye, Joseph: Thermodynamic Properties of Air. John Wiley & Sons., Inc., 1945.

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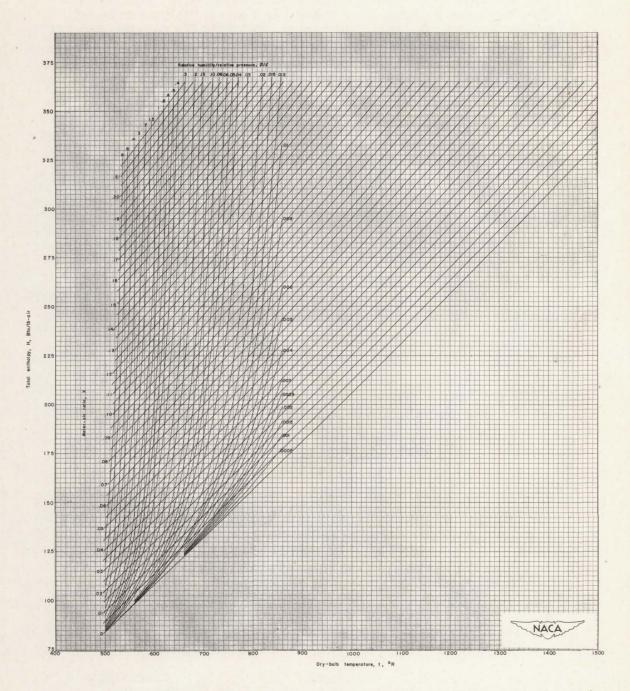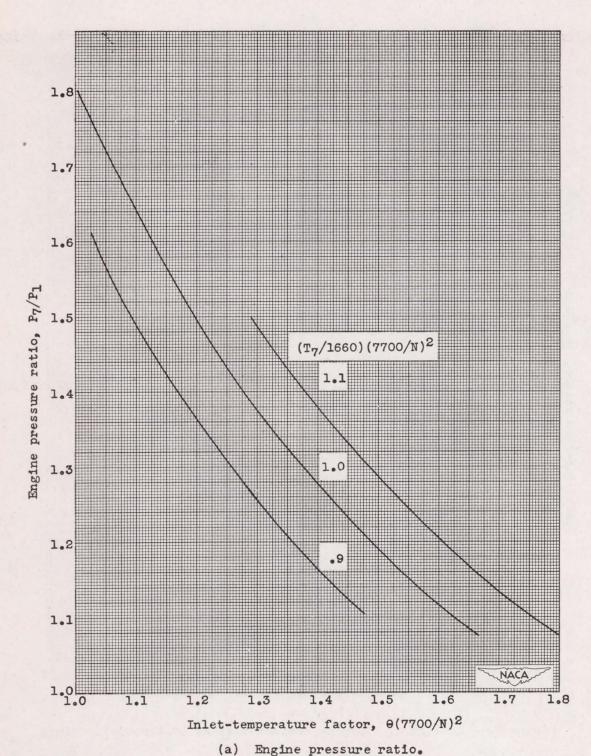
Figure 1. - Psychrometric chart. (A 17- by 22-in. print of this figure is attached.)

(a) Engine pressure ratio.

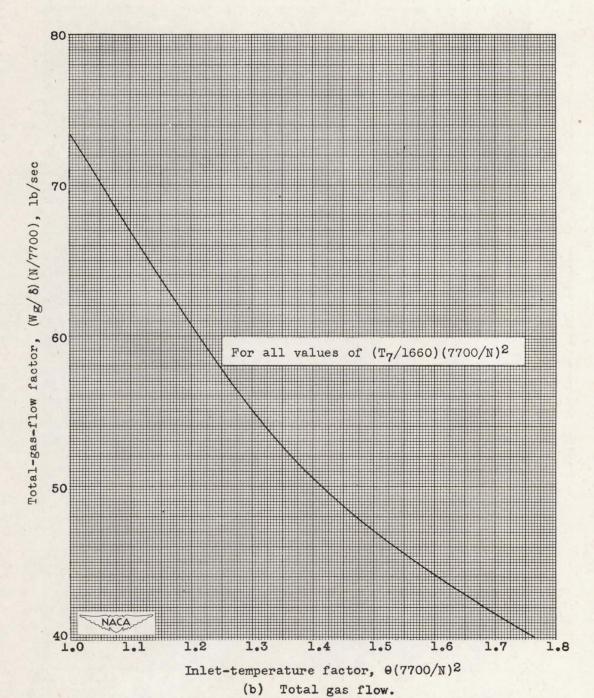Figure 2. - Pumping characteristics of turbojet engine.

(b) Total gas flow.
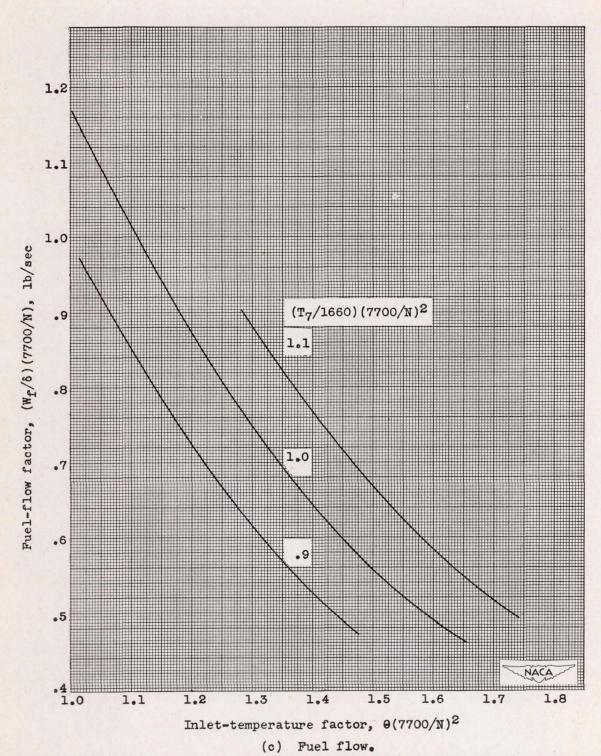
Figure 2. - Continued. Pumping characteristics of turbojet engine.

(c) Fuel flow.

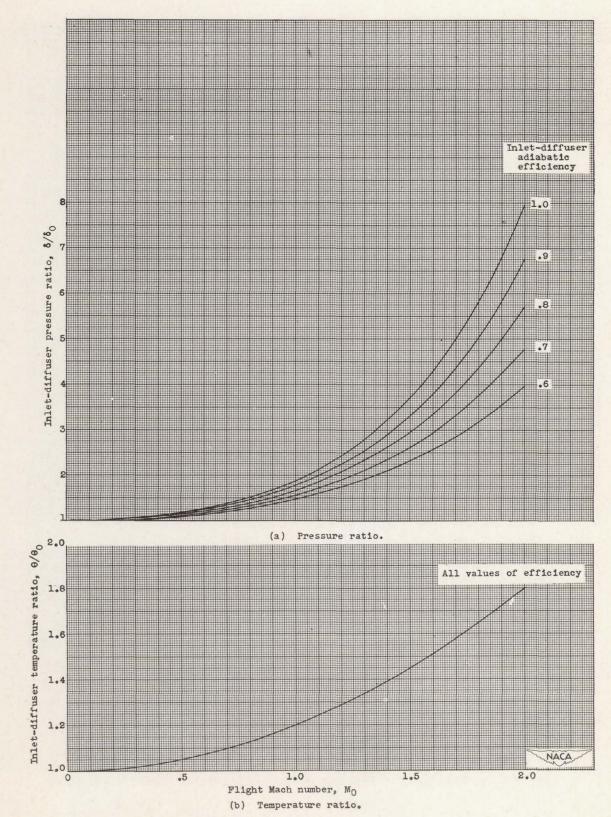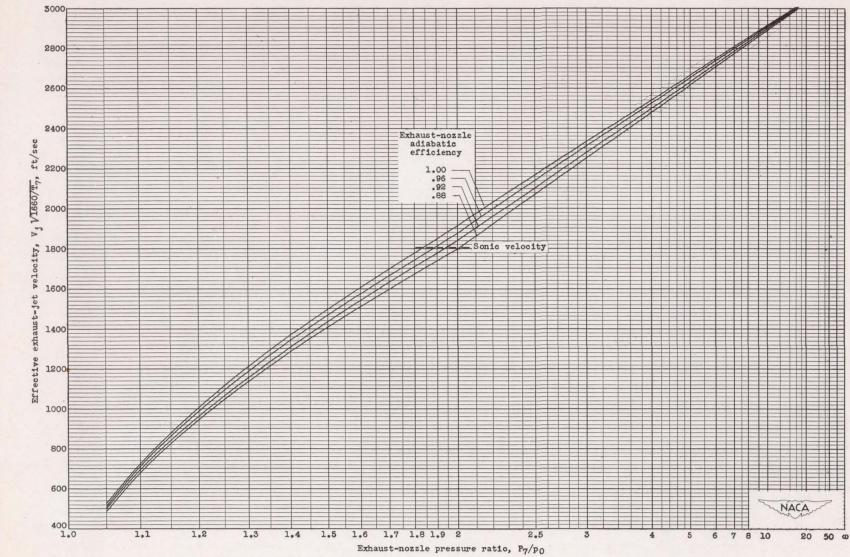Figure 2. - Concluded. Pumping characteristics of turbojet engine.

(a) Pressure ratio.

(b) Temperature ratio.

Figure 3. - Inlet-diffuser performance.

Figure 4. - Exhaust-nozzle performance.
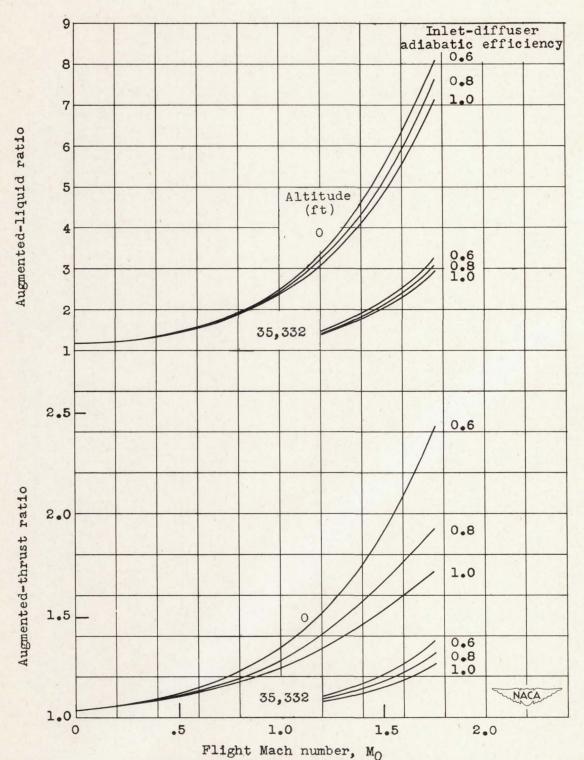
Figure 5. - Effect of flight Mach number, altitude, and inlet-diffuser
efficiency on augmented-thrust and augmented-liquid ratios.

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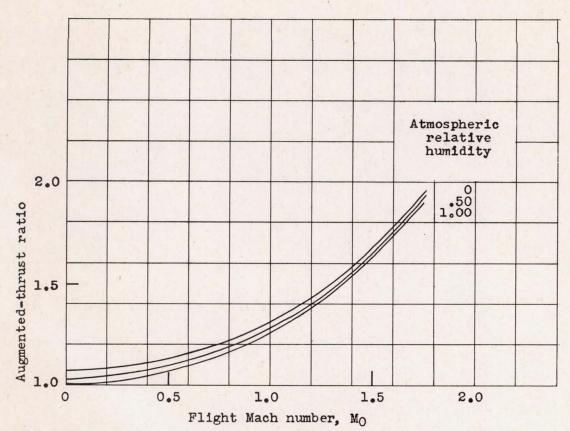
Figure 6. - Effect of atmospheric relative humidity and flight Mach
number on augmented-thrust ratio. Altitude, sea level; inlet-
diffuser efficiency, 0.8.
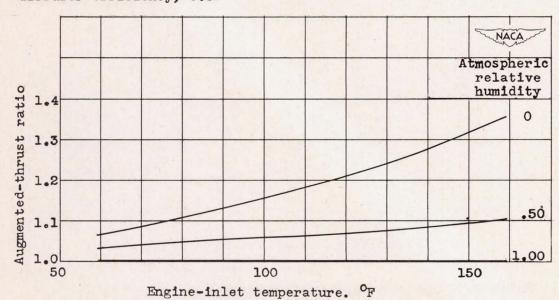
Figure 7. - Effect of engine-inlet temperature and atmospheric rela-
tive humidity on augmented-thrust ratio for sea-level, zero Mach
number conditions.

(a) Engine pressure ratio.
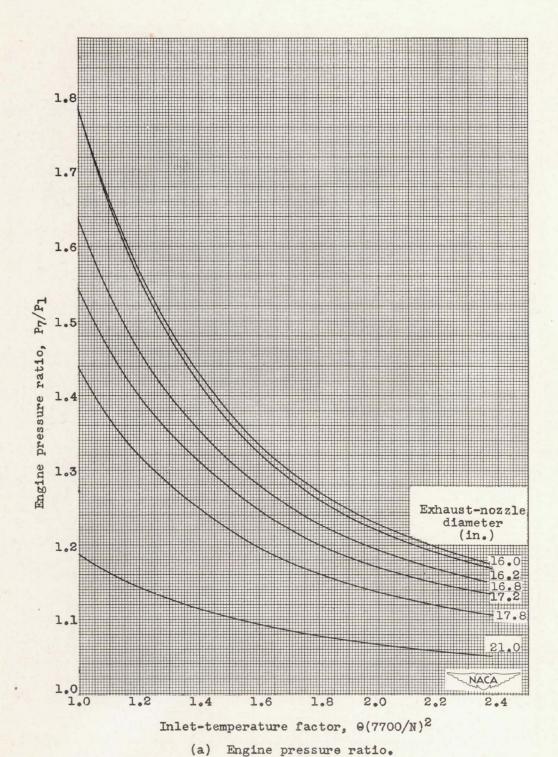
Figure 8. - Turbojet-engine performance for various exhaust-nozzle
diameters.

Inlet-temperature factor, $\Theta(7700/N)^2$

(b) Total gas flow.

Figure 8. - Continued. Turbojet-engine performance for various exhaust-nozzle diameters.

(c) Fuel flow.

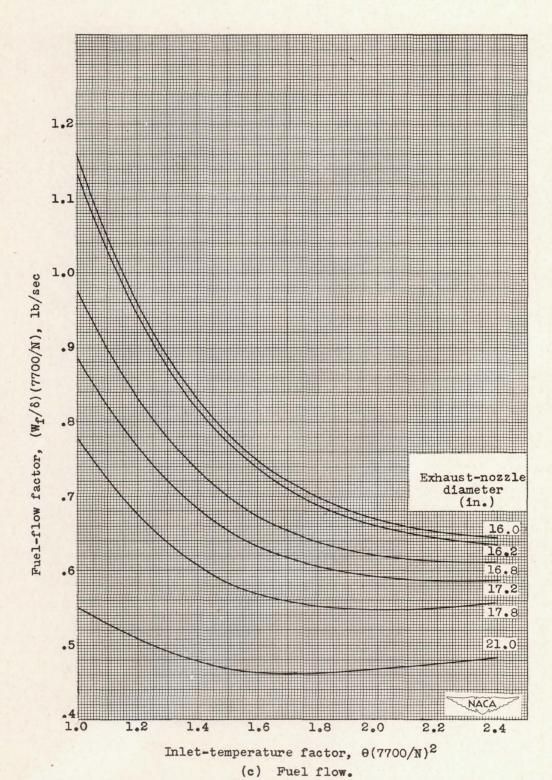Figure 8. - Continued. Turbojet-engine performance for various exhaust-nozzle diameters.
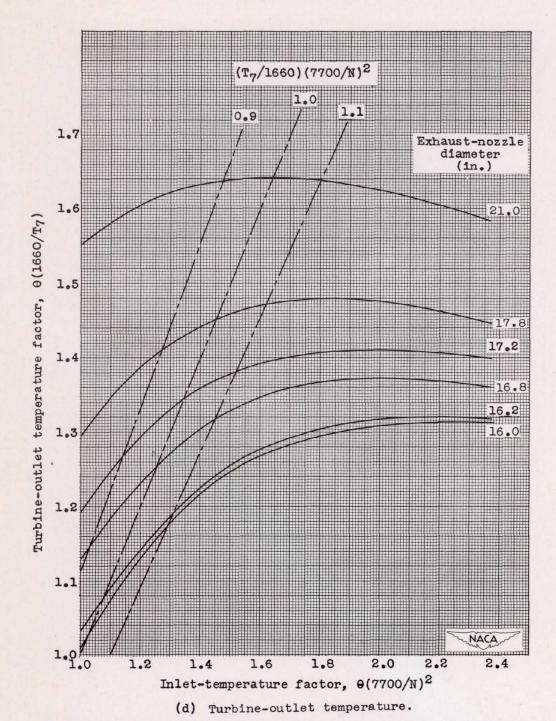
(d) Turbine-outlet temperature.
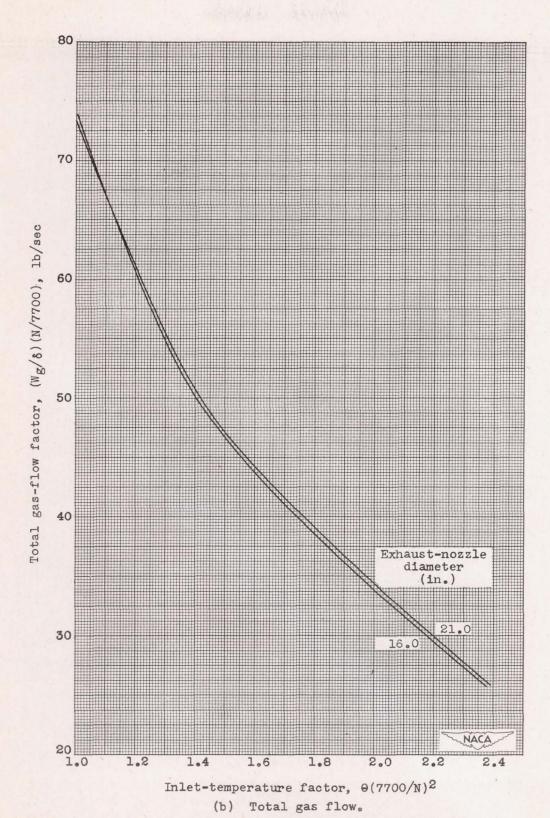
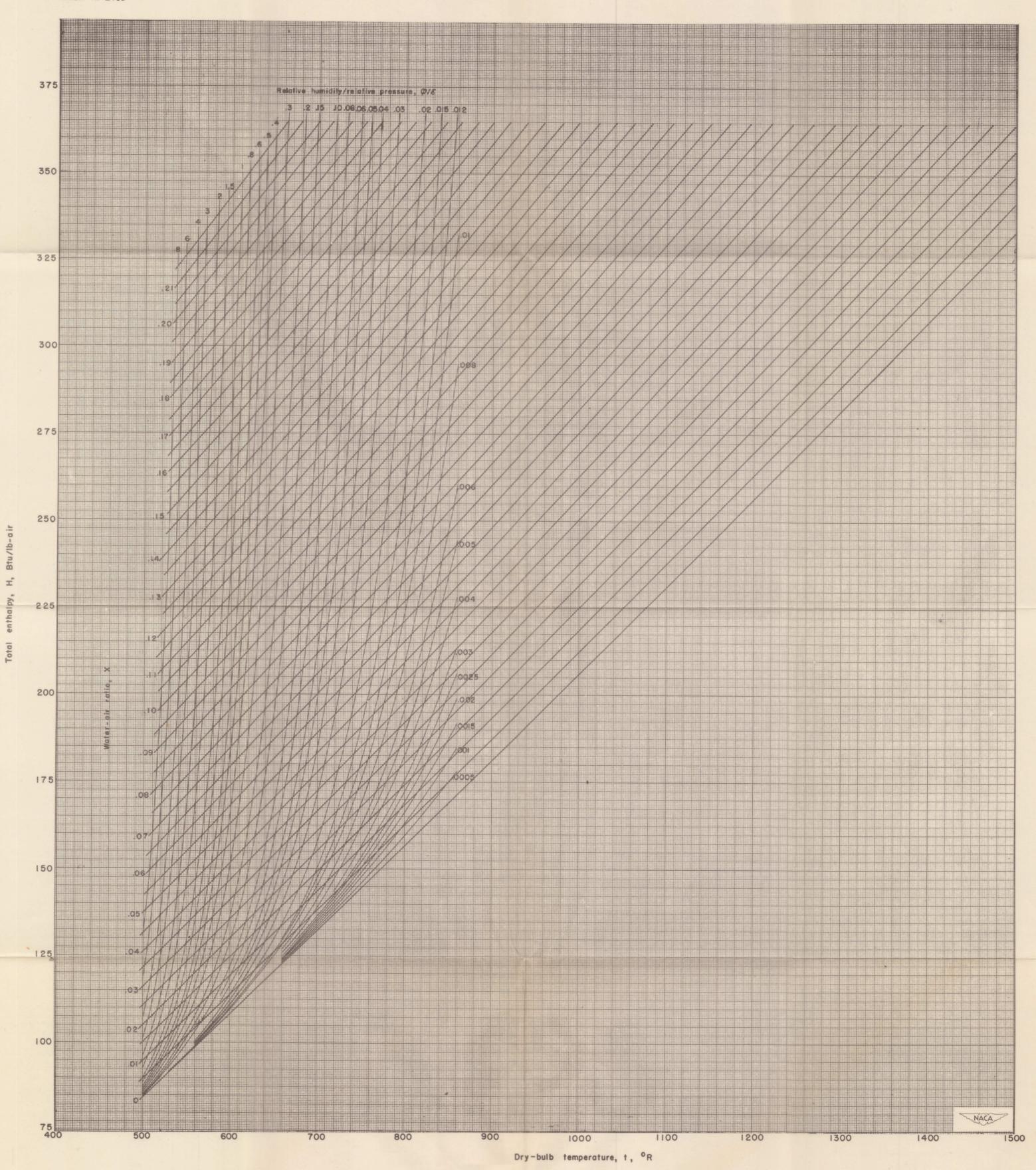Figure 8. - Concluded. Turbojet-engine performance for various
exhaust-nozzle diameters.

Figure I. - Psychrometric chart.